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Full-scale glare fuselage panel tests

R.W.A. Vercammen and H.H. Ottens

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R.W.A. Vercammen H.H. Ottens	970523	15	4

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ABSTRACT
In fuselage design studies there will always be the necessity to test components in a realistic way. The fuselage panel test facility at NLR offers the possibility to subject fuselage skin sections to residual strength and fatigue tests. The fatigue test loads simulate cabin pressurization in circumferential direction and axial loads representative of both cabin pressure and fuselage bending. To verify the test methodology and the specifications of the test set-up the performance is verified using a GLARE panel. The verification shows the suitability of the biaxial load introduction systems. In addition the GLARE fuselage panel tests proof that the use of GLARE leads to a substantial weight reduction without affecting the fatigue or static strength.

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FULL-SCALE GLARE FUSELAGE PANEL TESTS

Roland W.A. Vercammen*, Harold H. Ottens*

In fuselage design studies there will always be the necessity to test components in a realistic way. The fuselage panel test facility at NLR offers the possibility to subject fuselage skin sections to residual strength and fatigue tests. The fatigue test loads simulate cabin pressurization in circumferential direction and axial loads representative of both cabin pressure and fuselage bending. To verify the test methodology and the specifications of the test set-up the performance is verified using a GLARE panel. The verification shows the suitability of the biaxial load introduction systems. In addition the GLARE fuselage panel tests proof that the use of GLARE leads to a substantial weight reduction without affecting the fatigue or static strength.

INTRODUCTION

In service fuselages are subjected to cabin pressure and fuselage bending. Therefore it is highly desirable that in case of fuselage design studies curved structures are tested under those biaxial loading conditions. For this purpose barrel test set-ups are generally used. These set-ups, however, are less attractive for studies not directly related to a particular aircraft design: the radius of curvature is fixed, a large number of panels has to be tested simultaneously and the test frequency is rather low. In addition, barrel tests are expensive due to the large number of panels and long testing times. The panel test facility at NLR (fig.1) was developed to avoid these disadvantages. In this facility, which is flexible in panel diameter, panel width and panel length, a single fuselage panel can be tested in a relatively short time (De Jong (1)).

In the full-scale fuselage panel test facility at NLR fuselage panels, having curvatures between those of relatively small aircraft such as the Fokker 50 and those of large aircraft such as the Airbus A300, can be tested under flight loading conditions and can be subjected to residual strength tests. The loading sequences will be derived from the aircraft flight loading conditions. This results in circumferential load sequences caused by internal air pressure, synchronized with a longitudinal load spectrum representative of both cabin pressure and fuselage bending due to taxiing and gust loading. With an average number of 30 longitudinal load cycles and one cabin pressure load cycle within one flight, the testing frequency is about 10,000 flights per 24 hours. Next to the fatigue tests static strength and residual strength tests can be performed. During these tests large cracks such as a two-bay crack are allowed.

* National Aerospace Laboratory NLR, The Netherlands, Structures Department

In order to verify the full-scale fuselage panel test methodology the performance of the test set-up is checked using a Fokker 100 GLARE panel. This GLARE panel was supplied by Fokker Aircraft B.V.. Before testing the GLARE panel according to the commitments made by Fokker in the Brite Euram IMT 2040 project "Fibre reinforced metal laminates and CFRP fuselage concepts", NLR was allowed to use the Fokker 100 panel for several verification tests. The performance of the test set-up is checked by determining the radial expansion of the GLARE panel and the axial, tangential load introduction into the panel.

The fuselage panel was representative of the crown section just in front of the wing and had a GLARE A• skin and GLARE N stringers. GLARE A as skin material is weight favourable compared to GLARE C when the amount of necessary doublers for countersunk riveting is restricted to only doublers in the axial lap-joints. In the frame-skin attachments no doublers were required as the frames were connected to the stringers by means of cleats in stead of to the skin by means of castellations as is normally done by Fokker. Applying cleats led to a panel design with a threefold test objective:

- Verification of the applicability of GLARE A as a fuselage skin material for loading conditions which are representative of the crown section of the Fokker 100 fuselage.
- Generation of test evidence on the static strength and fatigue behaviour of rigid stringer-frame attachments in the GLARE fuselage.
- Determining the deterioration of the static strength of the GLARE skin after two times the design life (2*90,000 flights).

The overall panel dimensions are 1210 mm * 3030 mm containing five aluminium frames with a pitch of 500 mm, seven stringers with a pitch of 147 mm, seven stringers couplings and a longitudinal riveted lap-joint in the panel centre (fig.2).

One of the frames is a Z-shaped frame, the others are C-shaped frames. A complete top section made of GLARE A with GLARE N stringers and rigid stringer-frame attachments has a weight that is 63% of the current Fokker 100 design in aluminium (Wit (2)).

TEST FACILITY

The major components of the test facility are the main frame, the pressure chamber and the load introduction systems (fig.1). The main frame is a very stiff steel structure. It consists of heavy bottom and top beams and two vertical main columns. A pyramidal shaped frame, which houses the hydraulic actuator, is mounted above the top beam and two auxiliary vertical columns. The panel is mounted in the frame such that the centre of gravity of its cross-section is in the working line of the actuator. The height of the test facility is about 7.2m, the width is about 4.5m.

• GLARE A=GLARE 3-2/1-0.3: 2*(0.3mm 2024 sheet) + (0.25mm cross-ply glass prepreg)
GLARE N=GLARE 1-3/2-0.3: 3*(0.3mm 7475 sheet) + 2*(0.25mm UD glass prepreg)
GLARE C=GLARE 3-3/2-0.2: 3*(0.2mm 2024 sheet) + 2*(0.25mm cross-ply glass prepreg)

The pressure chamber is formed by a seal and base structure, connected to a transport system. The base structure of the pressure chamber is formed by a stiffened base plate and two support beams which are bolted to the vertical columns of the main structure. The base plate has a large central hole for air supply. At the front side the base plate has curved wooden blocks around the edges which form the side walls of the pressure chamber. An inflatable inner seal is mounted on the wooden blocks and the pressure chamber is closed by the panel. In order to accomplish an air-tight seal without net radial force acting on the panel edges, an inflatable outer seal is mounted at the outside of the panel just opposite the inner seal. The outer seal is bonded on the reaction frame, which consists of an open rectangular steel frame with curved wooden blocks. With the transport system the pressure chamber can easily be shifted aside during the test, which significantly improves the inspectability of the test panel. The chamber pressurization, axial loads and seals pressurization are regulated by a control system.

Pressurization of the fuselage panel is reacted to the test frame leading to tangential stresses in the skin and normal stresses in the frames. The ratio of the stresses in the skin and frames is determined and can be adjusted by the stiffness ratio of the skin-to-testframe and frame-to-testframe connections. The tangential stresses in the skin are taken out by bonded glass fibres. The loads in the frames are transferred to steel rods. Therefore the ends of the panel frames are locally reinforced. The wooden blocks have several holes through which the panel frame tensile rods are guided. The openings between the panel frame tensile rods and the hole edges are air-tight sealed with silicone rubber collars. In axial direction the panel-ends are loaded by rods which are connected to the panel-ends by steel brackets. At these axial panel-ends the stringers are ended and the stringers loads are taken by a gradually increased skin thickness (fig.2). This makes it possible to seal directly on the skin of the panel.

COMPROMISES

The philosophy of the full-scale fuselage panel test facility is that a curved testpanel should be loaded comparable to a panel in a pressurized fuselage subjected to bending. Therefore a radial expansion of the fuselage panel due to cabin pressure is necessary (fig.3). Pressurization of a fuselage results in the development of tangential stresses in the skin. The fuselage frames will prevent the free expansion of the skin and thus attract some load also. The stress levels in the frames will largely depend on the stiffness of the frame-skin connection. To obtain for all kind of frame-skin connections a radial expansion the ratio of stresses in the skin and frames must be correct and adjustable. Therefore NLR chose to use separate loading mechanisms for the skin and frames. The tangential stresses in the skin are taken out by bonded glass fibres. The loads in the frames are transferred to steel rods.

The steel frame loading rods bend due to the applied cabin pressure and the axial panel loads (fig.4). To keep these unwanted bending moments as small as possible, the diameter of the rods are minimized and the length is maximized. Due to cabin pressure, bending of the frame loading rods increased the frame stresses of the GLARE panel with 2%. When applying axial loads the axial elongation of the panel, and by that the bending moments, is different at all frame stations. In case of a flight these axial loads and cabin pressure are combined. Applying the severest flight of the Fokker 100 load spectrum the unwanted bending moments increase the frame stresses in the most upper frame of the GLARE panel with 5%. In case of static strength tests the applied axial loads are much higher. To limit the frame bending loads pre-stressing of the frames, by displacing the frame loading rods, is wanted.

It is possible to get rid of the unwanted frame bending moments, e.g. by using hinges. In that case however the frames will not be loaded in their working line.

The advantages of using unidirectional glass fibres is that the loads are very evenly introduced over the length of the panel (De Jong (3)). The GLARE panel proved that hardly any distance is required for stress redistribution, i.e. the tangential strain distribution is uniform after one stringer pitch from the panel edge (fig.5). In addition, the use of unidirectional fibres did not result in local stiffening of the panel edges in axial direction (fig.6). The length of the glass fibre sheets was chosen sufficiently large to limit the rotation of the fibres at the upper side of the panel owing to axial elongation of the panel. The small rotation that occur during an average flight alter not significantly the load transfer through the panel. In case of a static strength test the applied axial loads are much higher (fig.7). To limit the load transfer through the fibres a pre-displacement of the panel is wanted.

The glass fibres are bonded to steel tangential plates. Because of their large width the tangential plates act more or less as hinges. The outward movement of a panel due to pressurization is therefore nearly radially and will not result in a significant change in the shape of the panel from circular to oval. That the GLARE panel did not expanded pure radial (fig.8) is caused by the lap-joint and radius differences of the panel halfs.

In axial direction tensile loads simulate the fuselage bending. To introduce these loads into the skin and stringers a single load concept was chosen: At the axial panel-ends the stringers are ended and the stringers loads are taken by a gradually increased skin thickness. This reinforced skin is loaded by tensile rods. The option to introduce the loads into the skin and stiffeners by using unidirectional fibres and separate tensile rods would result in a complex loading system with two independent controlled axial load units. The GLARE panel proved that the chosen axial load introduction concept achieves a smooth distribution of the axial strain levels in the middle of the panel (fig.6,7) and that the influence of the stringer run-outs and skin reinforcements is negligible after the first frame (fig.10).

Conclusion: The unidirectional glass fibres introduce after one stringer pitch an uniform tangential strain distribution. In addition, during a fatigue test, the glass fibres do not disturb the uniform axial load introduction. Also the influence of the stringer run-outs and the skin reinforcements on the axial load introduction is negligible after the first frame. Finally, the frame loading rods cause small, unwanted bending moments in the frames.

GLARE PANEL LOADS

The GLARE panel is designed for the Fokker 100 loads in the crown section just in front of the wing. The axial loading sequences of the fatigue spectrum are derived from the spectrum applied in the Fokker 100 full scale test (Jongebreur (4)). The axial load is written as:

$$F_{axial} = a_1 * M_y + a_2 * \Delta p$$

with

F_{axial} = axial load in fuselage panel (N)
M_y = bending moment at particular Fuselage Station (Nm)
Δp = cabin pressure (N/m^2)
$a_{1,2}$ = Fuselage Station dependent constants

The spectrum consists of 36 repeating testblocks of 5000 flights. Each testblock of 5000 flights is subdivided in four subblocks of 1250 flights. Three subblocks are exactly equal, the fourth block is equal but for one severest flight. Within this spectrum eight flight types with different gust loading severity have been defined. Figure 9 shows the axial loading and frequency per 5000 flights for these typical eight flight types. Each flight has five segments: ground, initial climb, climb/descent, cruise and approach. During the ground segments the cabin pressure Δp=zero, during the cruise segment $\Delta p = \Delta p_{max}$. In case of climb/descent the cabin pressure variates between zero and Δp_{max}.

The fatigue test is followed by static tests. The GLARE panel is subjected to one Limit Load case, two Ultimate Load cases and a failure strength test. These static load cases are intended to demonstrate that after two times the design life (2*90,000 flights) and possible undetectable cracks in the GLARE skin the residual strength is still sufficient to carry Limit and Ultimate Load. The Limit Load case equals cabin pressure $\Delta p = \Delta p_{max}$ plus Limit Load bending moment. The first Ultimate Load case is the cabin pressure Ultimate Load case: $\Delta p = 2 * \Delta p_{max}$. The second Ultimate Load case equals 1.5*[cabin pressure $\Delta p = \Delta p_{max}$ plus Limit Load bending moment]. The second Ultimate Load case is followed by the failure strength test at $\Delta p = \Delta p_{max}$ by increasing the axial load until failure of the panel.

RESULTS GLARE PANEL TESTS

During the fatigue test the defined 180,000 flights were applied. The testing frequency was 10,000 flights per 24 hours. The axial loads had an accuracy of 0.5%, the chamber pressure reproduced within 3%. The panel was inspected by eddy current at the axial lap-joint, stringers run-outs and checked visually at the stringers couplings, doublers run-outs, stringers at the connection with the cleats, frames at the connection with the cleats and load introduction points of the frames. In accordance with the lifetime predictions (2) no cracks or damages were found.

The fatigue test was followed by static tests. The first static load case was the Limit Load test. For this load case figure 10 gives the axial strain levels from station zero till the fourth frame. Clearly visible is the pillowing of the skin. The lower axial strains in the second and third bay are caused by the stringers couplings and difference in stiffnesses of the frames. During this Limit Load test the panel extended linear, no permanent deformations took place. Some results of the first Ultimate Load case at 2*Δp, are visualized in figure 5. Figure 5 shows the tangential strain levels in the middle of the panel. The axial strain levels are less than 20% of the tangential levels. Like during the Limit Load test pillowing occurred and the panel expanded linear without permanent deformations. After the 2*Δp Ultimate Load case the second Ultimate Load test was applied. No final failure occurred and the strain distribution was, except for plasticity which occurred above Limit Load, conform the distribution during the Limit Load test. Conclusion: the Ultimate Load tests showed sufficient residual strength of the GLARE panel after two lifetimes fatigue loading.

To determine the deterioration of the panel after two lifetimes fatigue loading, the first few load steps of the Limit Load and Ultimate Load tests are compared with reference tests performed before the fatigue test. The deterioration of strain levels in axial direction was a small fixed value of 50 μstrain, caused by settling of the brackets, doublers and rivets. In tangential direction negligible differences ($\leq$ 1.0%) were to be noticed.

After the second Ultimate Load test, the cabin pressure was fixed at $\Delta p = \Delta p_{max}$ and the panel was axial loaded to failure. The axial strains are visualized in figure 11. Failure occurred at an axial load of 770 kN, i.e. 1.32* axial Ultimate Load. This axial load is 15% higher than the theoretically expected axial failure load (2). The skin failed between the third and fourth frame at the cross-section of the last rivets of the stringers couplings. Conclusion: in the crown section of the Fokker 100, GLARE A as fuselage skin material and GLARE N as stringer material are applicable. The skin, lap-joint and rigid stringer frame attachments (cleats) had enough fatigue and static strength.

CONCLUSIONS

The full-scale fuselage panel test facility has shown to operate according to the specifications: The biaxial load introduction systems proved to apply uniform loads in axial and tangential direction, due to bending of the frame loading rods the stress levels in the frame increased. In case of the GLARE panel uniform and realistic fatigue loads due to cabin pressure and fuselage bending could be applied at high frequency (10,000 flights per 24 hours). The conclusions with regard to the Fokker GLARE panel are:

- No cracks were found in lap-joint, cleats, stringers, skin, frames and stringers couplings after the two lifetimes fatigue test (inspection was done visually and by eddy current).
- The panel as a whole did not deform permanent at Limit Load.

- At both Ultimate Load testcases no failure occurred.
- Final failure occurred at 1.32 * axial Ultimate Load, which is 15% higher than theoretically expected.
- GLARE A can be used as fuselage skin material in the crown section of the Fokker 100. GLARE N can be applied as stringer material.
- The rigid stringer-frame attachments (cleats) are applicable in the crown section of the Fokker 100 fuselage.

Because of the excellent fatigue behaviour with sufficient static strength and a weight reduction of 37% the designed GLARE panel proved the feasibility of GLARE as fuselage material.

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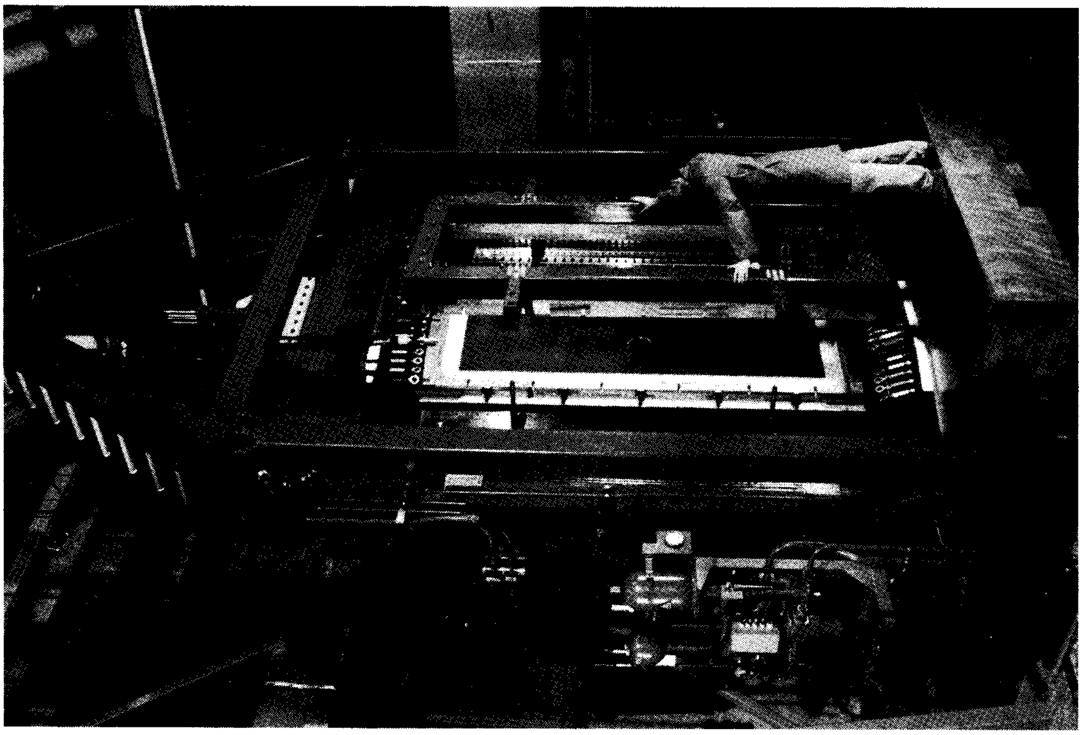

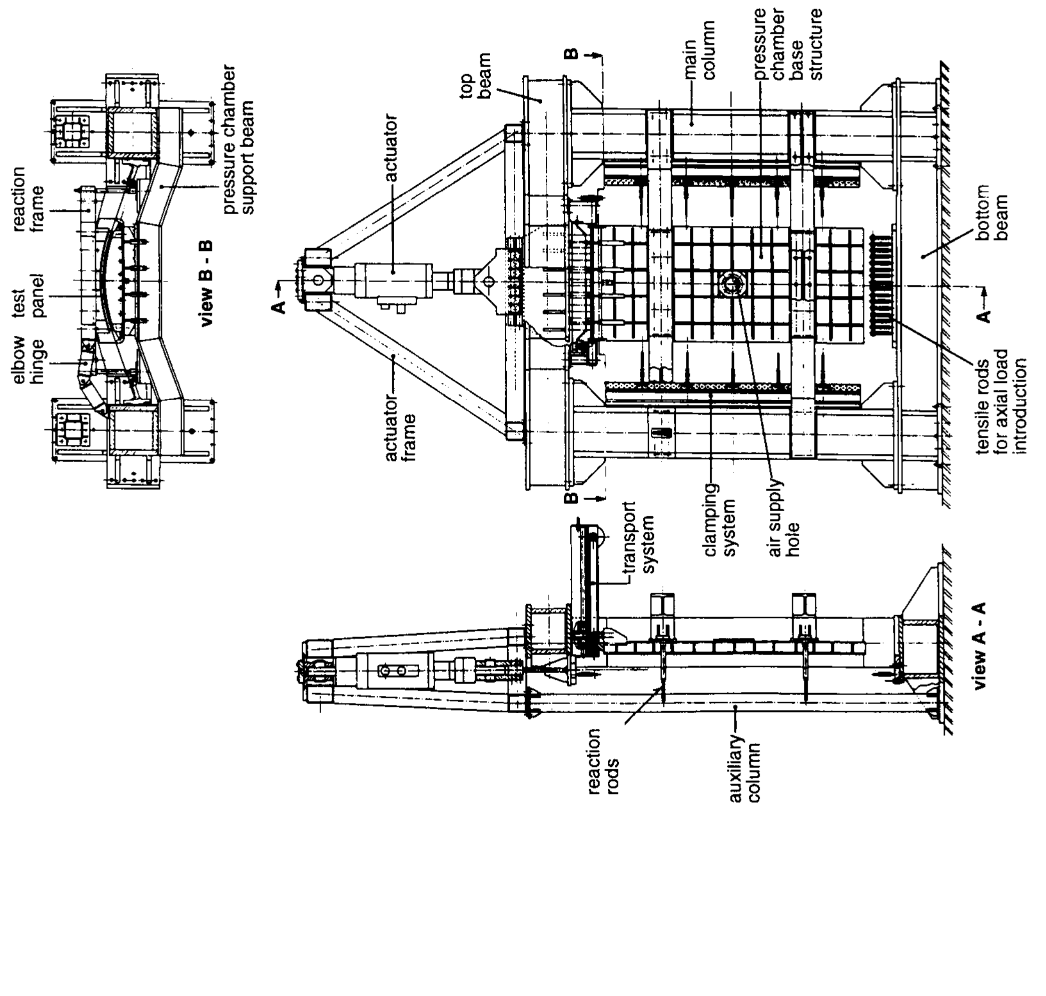

Figure 1 Fuselage panel test facility at NLR

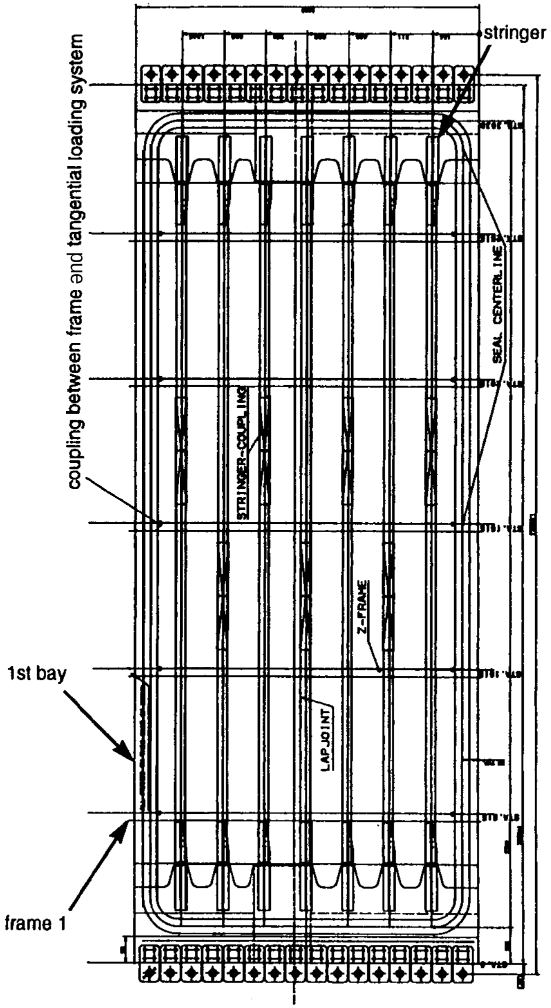

Figure 2 GLARE panel

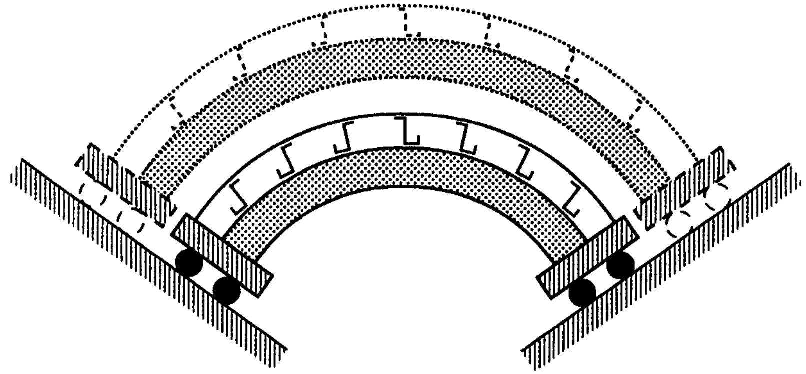

Figure 3 Ideal boundary conditions

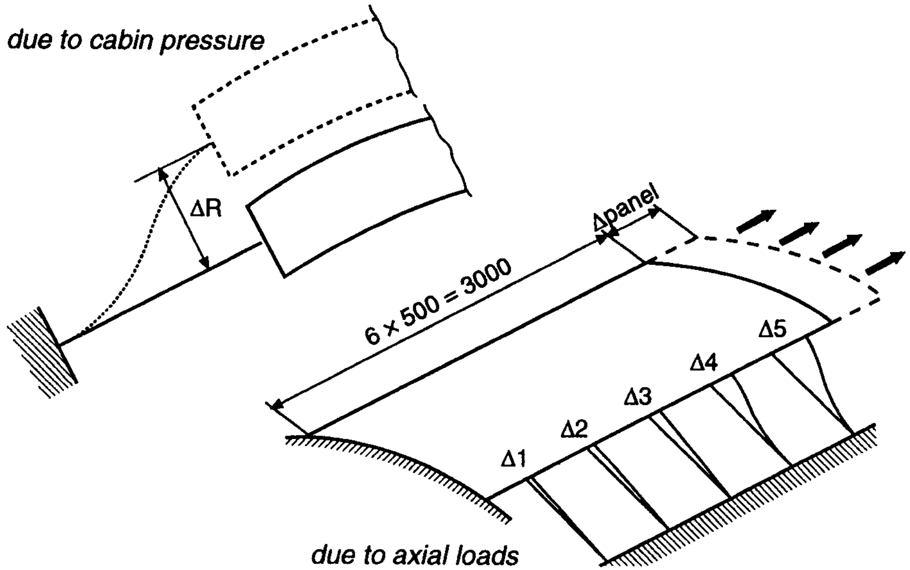

Figure 4 Bending of frame loading rods

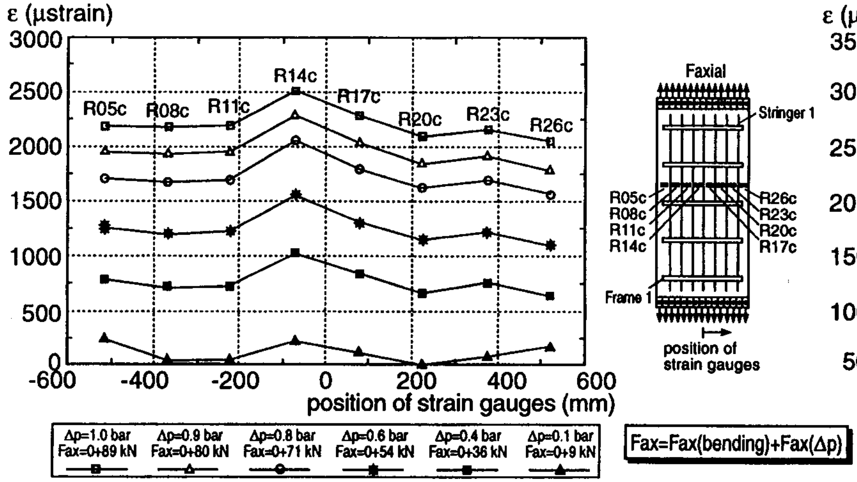

Figure 5 Tangential strains during 1st Ultimate Load test, in the middle of the panel

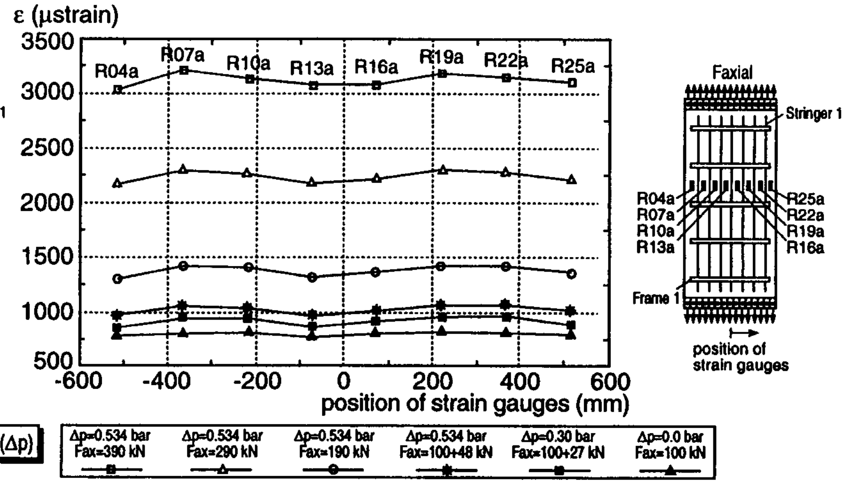

Figure 7 Axial strains during the Limit Load test, in the middle of the panel

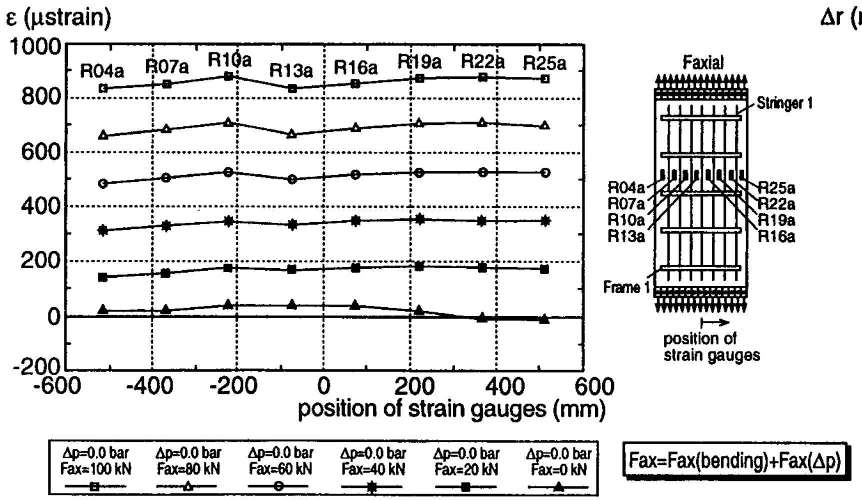

Figure 6 Axial strains due to axial loads

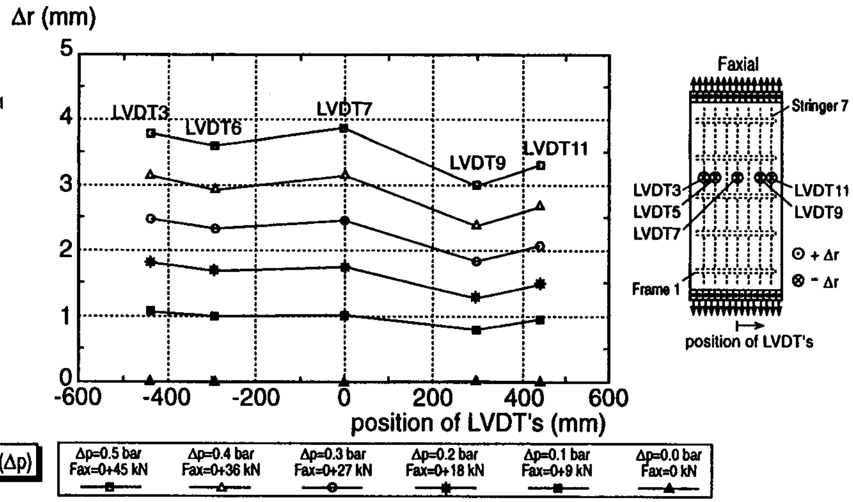

Figure 8 Radial expansion of the GLARE panel

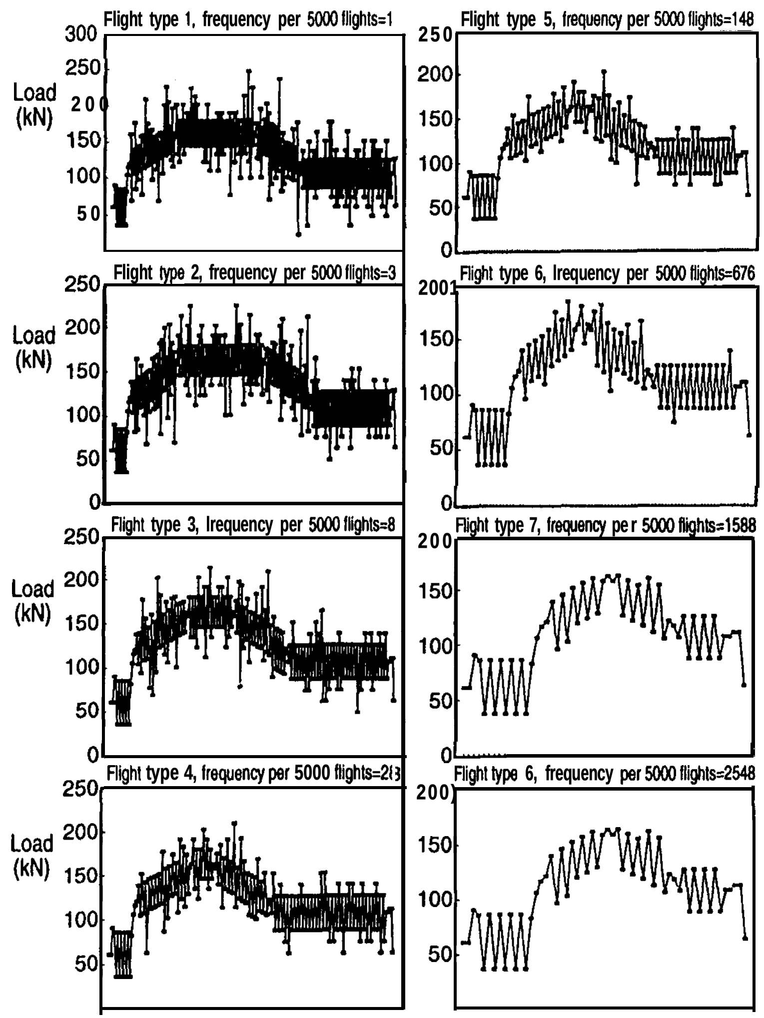

Figure 9 Axial load on the GLARE panel for the eight flight types

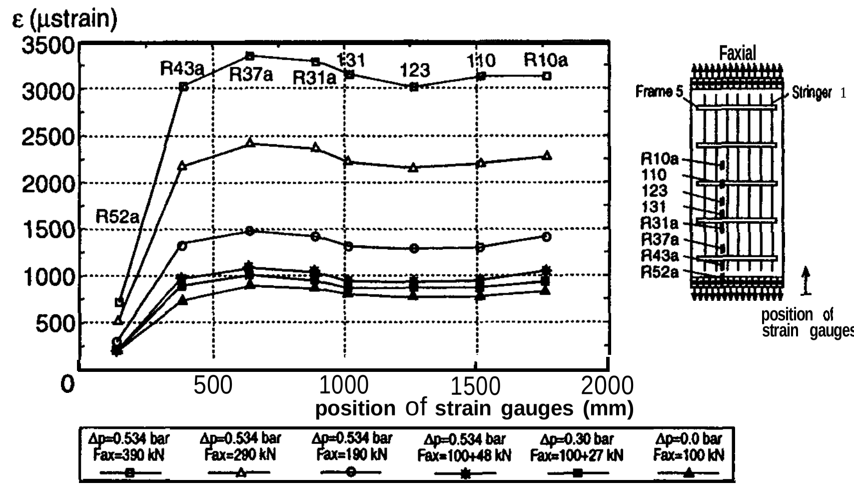

Figure 10 Axial strains during Limit Load test, between bottom and 4th frame

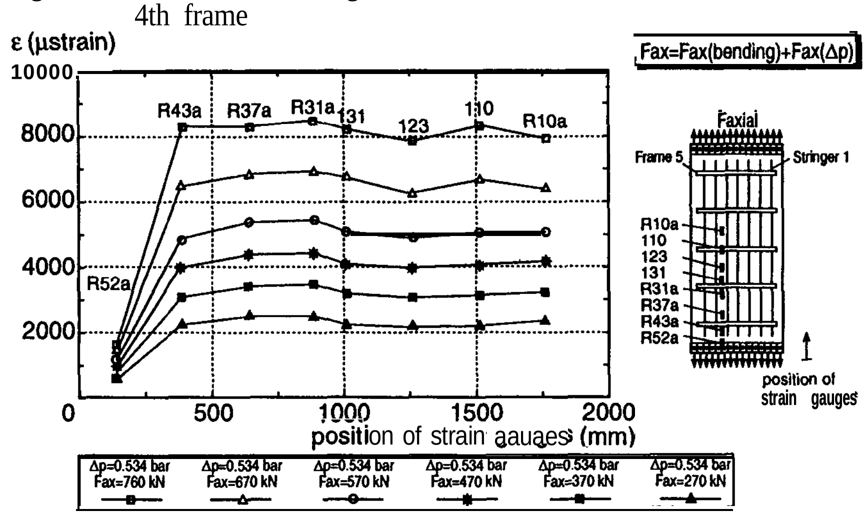

Figure 11 Axial strains during 2nd Ultimate Load test, between bottom and 4th frame